BELLCOMM, INC.
955 L'ENFANT PLAZA NORTH, S.W. WASHINGTON, D. C. 20024

SUBJECT: Spacecraft Payload and Mission Profile Changes for Lunar Exploration Missions - Case 310

DATE: March 17, 1969

FROM: D. R. Anselmo
J. L. Marshall, Jr.

MEMORANDUM FOR FILE

This memorandum describes Apollo vehicle sizing studies performed during the latter half of 1968 for lunar exploration missions. Early results of these studies were presented in August 1968 to the Apollo Program Office. These results have also served as the starting point for more recent efforts.

This study considered the modifications to the present Apollo system necessary to meet the objectives of a reasonable lunar exploration program. The study was directed toward extending lunar staytime and increasing payload, where payload is assumed to include increased consumables and habitability as well as scientific equipment and mobility aids. The resulting required changes to spacecraft weights and mission profiles will be discussed.

In this report minor changes from the August presentation were made to the CSM model. These changes were made to be consistent with a new but more meaningful treatment of CSM ΔV requirements. From a sampling of scientific sites extending over the front face of the moon, ΔV requirements were taken for the three best consecutive months in a year. The profiles employed varied depending on the accessibility of the site and included free return and hybrid missions both with and without DPS abort capability. An updated LM weight model was also included to reflect improved knowledge of consumables requirements and a more realistic approach to calculation of propellant requirements.

In very brief form, the selection of the LM weight model began with the determination of a lunar surface payload and a return payload consistent with LM performance capability. LM capability was extended beyond nominal Apollo by changes to the LM descent strategy beginning with a lower lunar parking orbit altitude and ending with a steeper descent flight path from high-gate to hover. Using the resulting LM model the CSM capability was then determined parametrically for different

(NASA-CR-103640) SPACECRAFT PAYLOAD AND MISSION PROFILE CHANGES FOR LUNAR EXPLORATION MISSIONS (Bellcomm, Inc.) 24 p

N79-7247?

Unclas
00/18 12695

(NASA CR OR TMX OR AD NUMBER) (CATEGORY)

B69 03056

BELLCOMM, INC.
955 L'ENFANT PLAZA NORTH, S.W. WASHINGTON, D. C. 20024

SUBJECT: Spacecraft Payload and Mission Profile Changes for Lunar Exploration Missions - Case 310

DATE: March 17, 1969

FROM: D. R. Anselmo
J. L. Marshall, Jr.

ABSTRACT

The feasibility of lunar exploration missions using Apollo hardware and mission profiles, with minimum required change to both, is investigated. The basic premises are that more payload both to the lunar surface and to lunar orbit will be required and that sites dispersed over the entire front face of the moon will be of specific interest.

The results show that spacecraft performance capabilities will allow missions to sites essentially over the entire face of the moon and further will allow a LM payload as high as 1800 lbs. to the surface concurrent with a CSM payload of 2700 lbs. to lunar orbit of which 700 lbs. may be returned to earth. The launch vehicle payload requirement which results is less than 105,000 lbs., which is within estimated capability.

The significant profile changes require the use of hybrid trajectory profiles and multi-impulse lunar orbit insertion and departure. It is suggested that the CSM deliver the LM to a 50,000 ft. lunar orbit and the CSM subsequently perform active rendezvous with the LM. In addition, modifications to the descent profile include lower high-gate altitude, steeper descent flight path, and reduced time from high-gate to touchdown.

CSM and lunar orbit payload weights. CSM models were then selected to be compatible with profile requirements for a comprehensive set of missions to sites of scientific interest.

For the spacecraft models selected the maximum required launch vehicle capability was near 105,000 lbs. Current launch vehicle payload estimates indicate that this requirement will not be a constraint.

Lunar Module Sizing

This section discusses in general terms the derivation of the baseline LM weights. For more detail on the assumptions made, see Appendix A.

In order to extend lunar staytime and increase payload without major LM hardware changes, it is necessary to modify the mission profile. For the nominal first lunar landing mission, the CSM has more propellant margin than the LM. Hence, one approach followed was to shift ΔV requirements from the LM to the CSM. Reducing the lunar parking orbit altitude from 60 n.m. to 50,000 ft. transfers about 150 fps of the LM descent requirement to the CSM. Similarly, the burden of the rendezvous could be transferred to the CSM, relaxing the LM ascent ΔV requirement (RCS) by the current budget allowance of 336 fps.*

The modified ascent strategy requires the ascent stage to insert into a 10 n.m. circular lunar orbit. In the nominal case this would place the LM in an orbit 10,000 ft. above the CSM which would then perform a transfer maneuver to rendezvous with the LM. The LM ascent budget presently provides 6030 fps for insertion into a 10 n.m. x 30 n.m. elliptic orbit. By using a 10 n.m. circular orbit this requirement can be reduced to 6000 fps. Including 21 fps for an out-of-plane correction of 0.5 degrees, the total nominal ascent budget becomes 6021 fps. The present dispersion allowance of 10 fps was used. The ascent budget employed here along with the present Apollo budget is given in Figure 1.

---

*With this profile the CSM would remain at 50,000 ft. until execution of rendezvous. The use of this low lunar parking orbit could present difficulties with thermal balance and line-of-sight rates to landmarks. If necessary, the CSM could place the LM in a 20 n.m. parking orbit and perform active rendezvous with the LM in a 30 n.m. parking orbit. The resulting LM ascent ΔV budget increase would be 60 fps and the landed payload penalty would be 176 lbs.

LM descent profile modifications and the resulting ΔV budget are discussed in detail in Reference 1. The ΔV budget is shown in Figure 1 and the profile modifications are summarized below.

- Reduce high-gate from 9650 feet to 6935 feet.
- Reduce time from high-gate to touchdown from 160 seconds to 120 seconds.
- Assume availability of high end throttling.
- Use 45° approach flight path angle.
- Increase redesignation and manual control allowance.

Two limitations on the maximum LM total weight are (1) the LM descent stage performance capability, including I_{sp}, ΔV, and propellant tank capacity, and (2) the landed weight limit, including the landing gear and other structural limitations. The curves in Figures 2, 3, and 4 indicate these limitations.

Figure 2 shows the performance limits. The maximum permissible LM Earth launch weight is plotted against the descent ΔV capability for three values of nominal descent engine I_{sp}. The curves assume the descent propellant tanks are loaded to their capacity. Clearly the higher I_{sp}'s (corresponding to more efficient engines) and lower ΔV requirements permit the highest LM weights. Note that on these curves an increase in LM Earth launch weight would result directly in an increased landed weight, since the propellant quantity is held constant.

Figure 3 shows how the maximum LM Earth launch weight is affected by the maximum allowable landed weight. Sets of curves are shown for two values of the landed weight limit, 16,200 pounds (current landing gear specification) and 17,000 pounds. On these curves, the landed weight is held constant while the usable propellant is allowed to vary. For a given I_{sp}, the lower ΔV's require less propellant, so the Earth launch weight (which is the sum of the landed weight, the descent propellant, and the descent expendables minus the CSM to LM transferred weight) is lower. If more propellant than the required amount were loaded (giving a higher Earth launch weight), all of it would not be consumed during the descent, so the landed weight limit would be exceeded. Also, since an engine with a higher I_{sp} requires less propellant for a given ΔV, the allowable Earth launch weight limit for a given landed weight and ΔV is lower for the higher I_{sp}'s.

Both the performance and landed weight limits are shown on Figure 4, which is a composite of the previous two figures using a descent I_{sp} of 301 seconds.* The intersection of these curves shows the maximum LM Earth launch weight and the corresponding ΔV capability, for the assumed propellant tank capacity and landed weight limit. For the descent ΔV budget shown in Figure 1 (6800 fps nominal, 150 fps dispersion) and an integrated average nominal descent I_{sp} of 301 seconds, Figure 4 indicates a LM Earth launch weight of 33,696 pounds and a landed weight of 16,797 pounds. Although the design specification landed weight is only 16,200 pounds, studies have indicated that the landing gear as designed is adequate for 17,000 pounds, and that only minor structural strengthening would be required for the heavier landed weight (Reference 2).

With the maximum LM Earth launch weight established, it is now necessary to determine the distribution of this weight between propellant, inert stage weights, and the allowance for payload. Since a reasonably stable reference was desired, the Apollo Program Specification control weights were chosen as a basis for the LM ascent and descent inert weights. However, it should be noted that although the latest reported (Reference 10) LM-6 ascent weight is 68 pounds below the control value, the descent stage exceeds its control weight by 148 pounds.

Since the descent propellant was specified in determining the total LM Earth launch weight, the remaining variables are the payload and the ascent propellant. Figure 5 shows the tradeoff between payload carried to the surface and payload returned to lunar orbit. The middle curve corresponds to the I_{sp} and LM weight discussed above (301 seconds nominal and 33,696 pounds). Two propellant limit lines are shown: one representing the capacity of the main ascent propulsion system (APS) propellant tanks plus RCS propellant available for translational ΔV during ascent, and one representing the APS

---

*Reference 3 uses a nominal descent I_{sp} of 300.5 seconds, with a dispersion of $\pm$3.98 seconds.

propellant capacity. Points lying to the right of the APS line require more propellant than the APS tank limit, and points to the right of the APS + RCS line require more propellant than the APS tank limit plus RCS propellant available for translational ΔV. Points to the left of these lines require off-loading either RCS or APS propellant, or both. The flat portions of the curves (for return payloads less than 220 pounds) result from the current practice of basing ascent propellant loading on the touchdown abort condition. The assumptions used for consumable, transferred, and jettisoned items are such that the lunar lift-off weight for a nominal mission (excluding science allowance) is 220 pounds lighter than the touchdown abort weight. Thus, on a nominal mission, any return payload up to 220 pounds would not require additional ascent propellant, or in terms of Figure 5, would not require a corresponding change in surface payload. Since 220 pounds of return payload is considered adequate, the allowable surface payload is 1854 pounds. The surface payload weight in Figure 5 includes the current 300 pound allowance for science. It should be noted that since the curve of Figure 5 is dependent upon the assumed set of ascent and descent stage inert weights, the surface and/or return payload must be adjusted to account for any differences between the assumed inert stage weights and the actual flight weights.

A summary of the LM weights, plus the performance values assumed, is given in Figure 6.

Command and Service Module Sizing

Command and Service Module performance analysis is dependent upon the Lunar Module model. For CSM considerations the 33,696 lb. LM weight model developed in the previous section will be employed. A convenient graphical representation for CSM capabilities is illustrated in Figure 7. Here ΔV capability with the LM attached is plotted versus ΔV capability without the LM. This plot is useful for comparing mission ΔV requirements with the capability of a given configuration since various mission profiles have different distributions of ΔV with and without the LM attached. For the spacecraft weight models given the line on this graph represents the combinations of ΔV which require all usable propellant.

Specifically, the line on Figure 7 shows the SPS capability with a LM weight of 33,696 lbs. which was derived for an 1854 lb. landed payload and a 220 lb. return payload. Three different CSM models were compared, each of which results in equal performance capability. For each model the translunar and lunar orbit expendables were 300 and 700 lbs.

respectively. The CSM inert weight used is the control weight value. A 200 lb. increase in lunar orbit expendables was made to provide for longer lunar orbit staytimes. Current CSM inert weight is 866 lbs. below the control weight value (Reference 10). Three different lunar payload configurations were considered. These are designated as model numbers I, II and III in Figure 8. Note that both payload jettisoned in lunar orbit and payload taken to lunar orbit and returned to earth are considered.

ΔV requirements for various missions can be superimposed on this plot. A point lying below the line represents a mission that is feasible, and the ΔV margin can be read off directly. Note that any point on the line represents an equivalent CSM ΔV budget since all points on the line give equal propellant consumption.

Mission ΔV requirements for lunar exploration will of course depend on the launch date, the landing site and the mission profile selected. Possible profiles could employ free return, non-free return, or hybrid trajectories. For landing sites far off of the lunar equator a three impulse lunar orbit insertion and departure will be required in some cases. The shaded area in Figure 7 delineates the ΔV requirements to reach sites of scientific interest. The envelope was constructed to enclose the ΔV requirements for missions to each site considered for three consecutive months. For less accessible sites the hybrid profile was used and in some cases a three impulse LOI and TEI was necessary. A series of reports prepared by TRW Systems for MSC supplied most of the ΔV data for these sites (References 5-9). The requirements include costs for CSM active rendezvous with the LM and standard contingency and dispersion allowances as well as delivery of the LM to a 50,000 ft. circular orbit. The CSM mission independent ΔV budget and consumables are discussed in Appendix B. Figure 9 lists the sites and profile types which were employed in the generation of the ΔV requirement envelope. A ΔV capability of 3220 fps with the LM attached and 4000 fps without the LM would assure accessibility of all sites considered for at least three consecutive months.

Summary

LM payload capability was increased by shifting ΔV requirements from the LM to the CSM (reducing the lunar parking orbit altitude from 60 n.m. to 50,000 ft. and employing CSM active rendezvous) and by modifying the LM descent profile (reducing high-gate altitude, reducing time from high-gate to touchdown, and using 45° approach flight path angle). For the descent, a nominal ΔV of 6800 fps with a dispersion allowance of 150 fps was used. For the ascent, the nominal ΔV was 6021 fps with a dispersion allowance of 10 fps. The specific impulse values for descent and ascent were 301 $\pm$4 sec. and 307.5 $\pm$4.1 sec. respectively. The resulting total LM Earth launch weight was 33,696 lbs., with a lunar surface payload of 1854 lbs. (including the amount required for increased staytime) and a return payload of 220 lbs.

The profile changes affecting CSM performance requirements are the use of other than free return trajectories, delivery of the LM to a lower parking orbit altitude, and CSM active rendezvous. Using these profile changes, missions are feasible to all of the science sites studied for at least three consecutive months. A total ΔV budget for the CSM, including the mission independent velocity requirements, is composed of 3220 feet per second for maneuvers with the LM attached and 4000 feet per second for maneuvers without the LM. Performance consistent with this budget provides accessibility to all of the sites studied.

It was shown that 2700 lbs. of CSM payload could be delivered to lunar orbit, leaving the LM and 2000 lbs. in lunar orbit and returning 700 lbs. of payload to earth. The launch vehicle payload requirements in this case are less than 105,000 lbs.

D R Anselmo

D. R. Anselmo

James L. Marshall, Jr.

2013-$\frac{\text{DRA}}{\text{JLM}}$-srb J. L. Marshall, Jr.

Attachments

REFERENCES

1. Heap, F., and V. S. Mummert, "LM Descent Profiles with Steep Final Approach Phases", Bellcomm Memorandum for File, March 17, 1969.

2. "Apollo Applications Program Study - Utilization of LM for Advanced Applications", Grumman Aircraft Engineering Corp., Report No. ARP 325-3, December 15, 1967.

3. "Supplement I to Apollo Spacecraft Weight and Mission Performance Definition (December 12, 1967)", NASA MSC, G. M. Low, July 10, 1968.

4. "Consumables Analysis for the Apollo Mission G Reference Trajectory", NASA MSC Internal Note No. 68-FM-246, October 10, 1968.

5. "Evaluation of the Three Impulse Hybrid Trajectory for Missions to Discrete Lunar Science Sites", TRW Letter 3421.5-120, September 13, 1968.

6. "Revision to the Results of Advanced Lunar Mission Targeting to the Crater Abulfeda in 1971", TRW Letter 3421.5-125, September 16, 1968.

7. "Results of Advanced Lunar Mission Targeting to the Crater Littrow in 1971", TRW Letter 3421.5-121, September 23, 1968.

8. "Results of Advanced Lunar Mission Targeting to Crater Hyginus in 1971", TRW Letter 3421.5-155, November 4, 1968.

9. "Results of Advanced Lunar Mission Targeting to Crater Dionysius in 1971", TRW Letter 3421.5-174, December 12, 1968.

10. "Apollo Spacecraft Configuration, Weight and Performance Summary", MSC, February 1969.

APPENDIX A

LM Weight and Performance Model

The major assumptions on the mission profile were discussed in the body of this memorandum. This appendix covers additional assumptions used in deriving the figures.

The approach used is the one presented in Reference 3. However, values for propellant capabilities, consumables, transferred items and jettisoned items were updated to reflect profile differences and more current data. Some of the more important features are outlined below:

1. Previous reporting on Apollo Spacecraft has allowed simultaneous occurrence of minimum I_{sp}, maximum ΔV, maximum propellant residuals, etc. The approach used here combines independent dispersions statistically (root-sum-squared) rather than adding them directly.

2. Previous calculations have generally made the conservative assumption that consumable items (e.g., RCS attitude control propellant) used during a burn are not consumed until the completion of the burn. The approach now used assumes that all items consumed during the powered descent are used linearly as a function of time.

3. The current control weights (ascent = 4950 inert plus 570 usable RCS propellant, descent = 4750) include all unusable propellants. Part of these residual propellants are considered dispersions, and since they are independent of the other dispersions mentioned (I_{sp}, ΔV, etc.), they can be combined statistically with the others. However, when this is done, the control weights should be adjusted accordingly. The equivalents to the ascent and descent control weights are 4867.1 and 4551 pounds respectively. This descent value includes the current 300 pound allowance for science payload; it was more convenient to consider this as part of the overall payload, so the descent inert weight was decreased to 4251 pounds.

4. The RCS propellant budget is based on Reference 4, modified to reflect mission profile changes. A total of 282.5 pounds was assumed for attitude

control and checkout, including LM active docking. As shown in Table A-1, this leaves 288.3 pounds available for translational ΔV; however, for the reference mission chosen, this 288.3 pounds and some of the ascent propellant were off-loaded.

5. Table A-1 also indicates the ascent and descent propellant tank capacities. The items listed in this table are defined in Reference 3.

6. For the reference mission chosen, the distribution of the weight at several points in the mission is shown in Table A-2.

TABLE A-1

LM Propellant Capacities

	RCS
Deliverable	570.8
Mixture Ratio and Gauging Inaccuracy	-81.0
Nominal Requirement	-156.9
Dispersions Allowance	-44.6
Available for ΔV	288.3

	Descent	Ascent
Usable*	17,698.2	5,099.8
Malfunction Allowance	-110.5	-11.1
Performance Uncertainty Allowance	-113.0	--
Dispersions Allowance	-308.8	-60.2
RCS Propellant Available for ΔV	--	+288.3
Total Usable for ΔV	17,165.9	5,316.8

*Does not include increase that would result from using the "stretched tanks" loading concept.

TABLE A-2

LM MISSION WEIGHT HISTORY

EVENT	WEIGHTS						
	ASCENT INERT	RCS PROPELLANT	ASCENT PROPELLANT	DESCENT INERT	DESCENT PROPELLANT	PAYLOAD	TOTAL LM
EARTH LAUNCH	4867.1	282.5	4743.1	4251.0	17698.2	1853.8	33695.7
CREW; TRANSFERRED ITEMS; CONSUMABLES	+463.9	-5.2	-	-	-	-	+458.7
SEPARATION FROM CSM	5331.0	277.3	4743.1	4251.0	17698.2	1853.8	34154.4
CONSUMABLES	-	-25.7	-	-	-	-	-25.7
START DESCENT	5331.0	251.6	4743.1	4251.0	17698.2	1853.8	34128.7
DPS; CONSUMABLES (USED LINEARLY)	-	-77.8	-	-88.2	-17165.9	-	-17331.9
TOUCHDOWN	5331.0	173.8	4743.1	4162.8	532.3	1853.8	16796.8
CONSUMABLES; ITEMS TRANSFERRED, JETTISONED, & PICKED UP FROM SURFACE	-210.0	-10.0	-	-4162.8	-532.3	-1633.8	-6548.9
LUNAR LIFT-OFF	5121.0	163.8	4743.1	-	-	220.0	10247.9
APS; CONSUMABLES (DROPPED AFTER BURN)	-40.9	-38.2	-4671.8	-	-	-	-4750.9
HARD DOCKING	5080.1	125.6	71.3	-	-	220.0	5497.0

APPENDIX B

CSM Weight and Performance Model

Because of the mission dependent nature of the ΔV requirements the CSM analysis was simplified to consider velocity changes in two categories: (1) with the LM attached and (2) without the LM attached. Since maneuvers made in the lunar vicinity such as CSM plane change and CSM active rendezvous are made with approximately 500 lbs. of crew and equipment removed, the model is inherently conservative. Consumables were broken into two categories: (1) weight loss before LOI and (2) weight loss in lunar orbit. The translunar consumables were dropped entirely before calculation of ΔV capability with the LM and the lunar orbit consumables were dropped before calculation of ΔV capability without the LM. The translunar and the lunar orbit consumables were 300 and 700 lbs. respectively. Table B-1 gives a breakdown of the mission consumables employed.

The mission independent CSM ΔV budget provides for delivery of the LM to a 50,000 ft. parking orbit and subsequent CSM active rendezvous. The delivery to a 50,000 ft. circular orbit by a Hohmann maneuver requires 150 fps additional CSM ΔV. For LM rescue by CSM active rendezvous the present budget provides 790 fps which was retained for this study as the maximum required rendezvous budget. The translunar and transearth midcourse correction budgets were set at 100 fps and 60 fps respectively. The contingency allowance, including any orbit return, was set at 210 fps. The mission independent CSM ΔV budget is given in Table B-2. An SPS specific impulse of 311 seconds was employed throughout the study.

TABLE B-1

CSM Expendables

Translunar		
CSM Inert	- 91	
RCS	-209	
	-300	
Total translunar		-300
Lunar Orbit		
LOI to Separation		
CSM Inert	- 36	
RCS	- 56	
	- 92	
Separation to Rendezvous Start		
CSM Inert	- 55	
RCS	-134	
	-189	
Rendezvous Start to TEI		
CSM Inert	-289	
RCS*	-130	
	-419	
Total lunar orbit		-700

*Allows for a nominal rendezvous with the CSM in a 50,000 ft. orbit and the LM in a 10 n.m. (60760 ft.) orbit. The worst case rescue requirement would be on the order of 600 lbs. of RCS which is carried through TEI in this model.

TABLE B-2

Mission Independent CSM ΔV Budget

<u>ΔV with LM</u>

Midcourse Corrections	100
LM to 50,000 feet	150
	250 fps

<u>ΔV without LM</u>

CSM Active Rendezvous	790
Midcourse Corrections	60
Contingencies	210
	1,060 fps

LM ΔV Budget

	Apollo Present	LEP Proposed
Ascent		
Nominal	6050	6021
Dispersions	10	10
	6060	6031
RCS Rendezvous	336	-
Descent		
DOI	71	-
Braking	5345	5655
Approach	866	717
Landing	406	56
	6688	6428
Redesignation, Manual Control	335	372
Dispersions	157	150
	7180 fps	6950 fps

FIGURE 1

LM VEHICLE SIZING
(PERFORMANCE LIMITS)

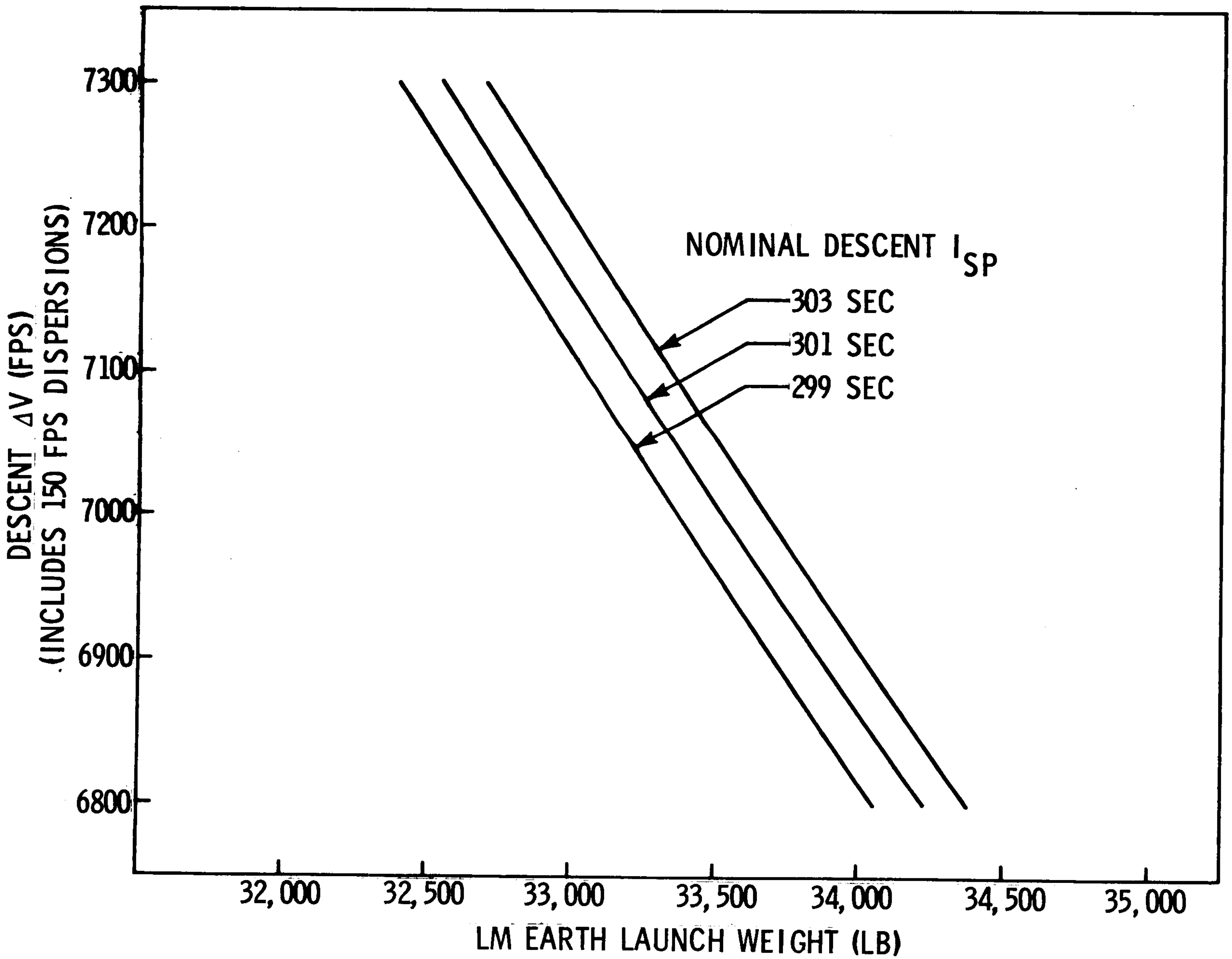

FIGURE 2

LM VEHICLE SIZING
(LANDED WEIGHT LIMITS)

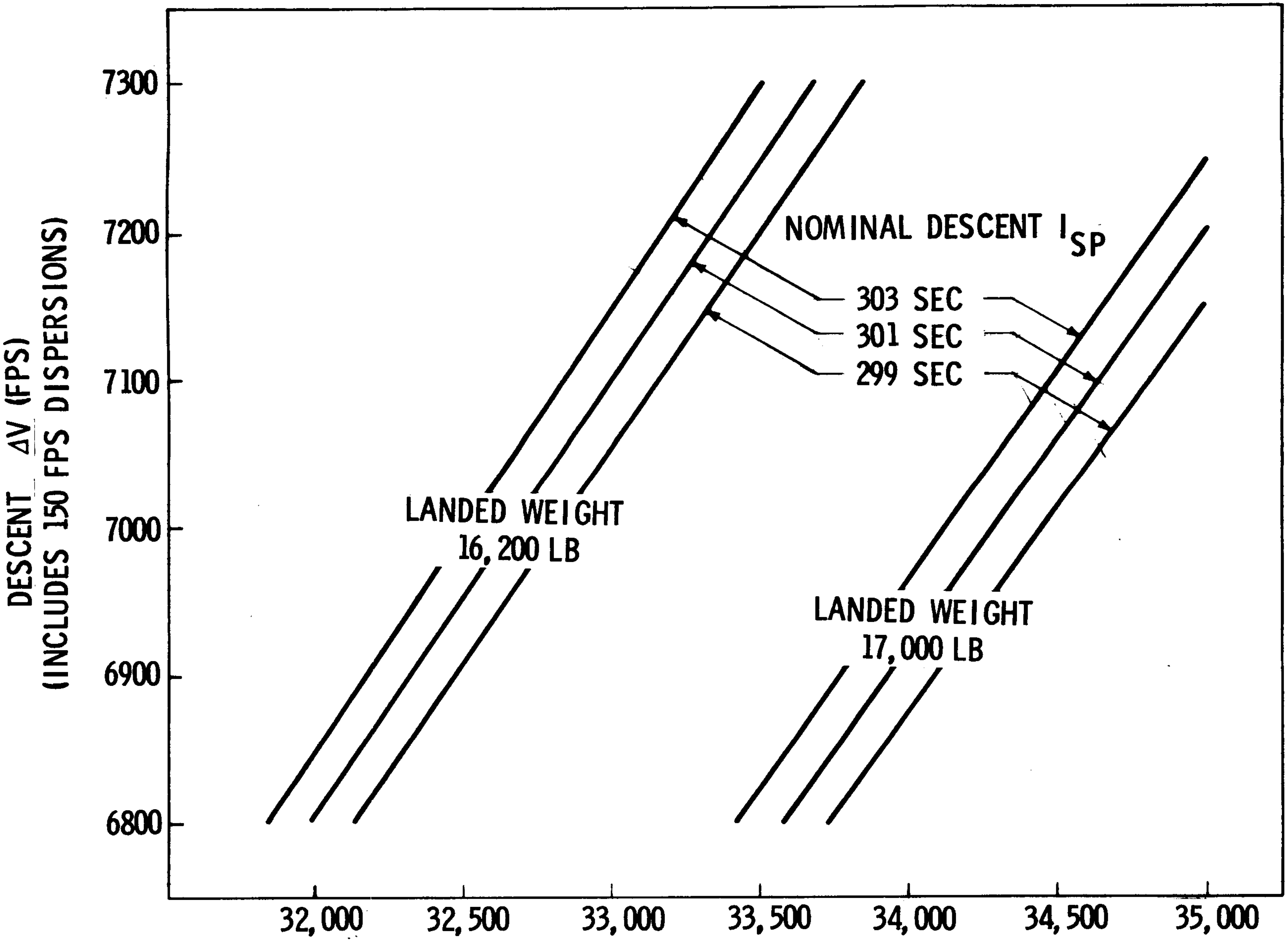

FIGURE 3

LM VEHICLE SIZING

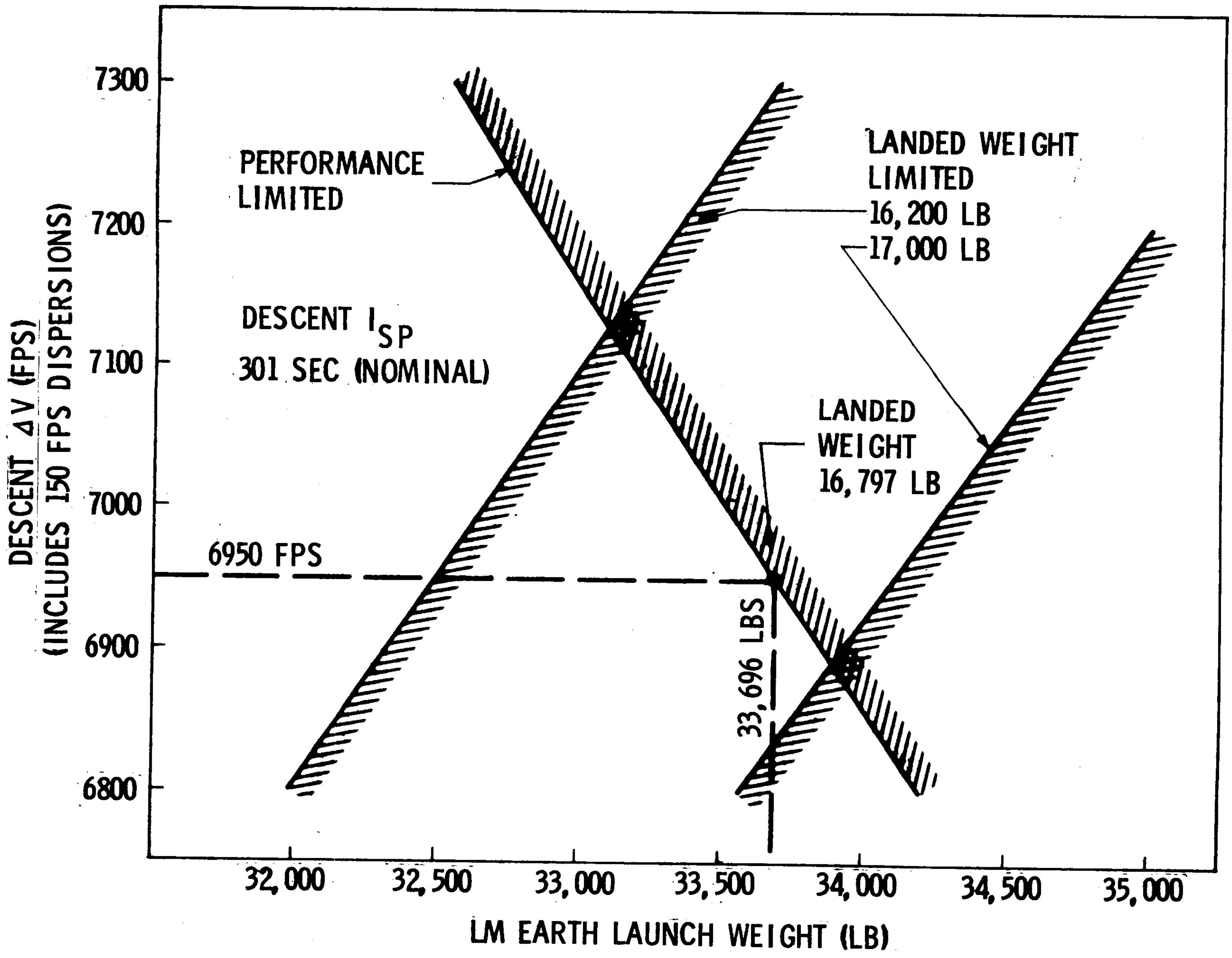

FIGURE 4

LUNAR MODULE PAYLOAD TRADE-OFFS

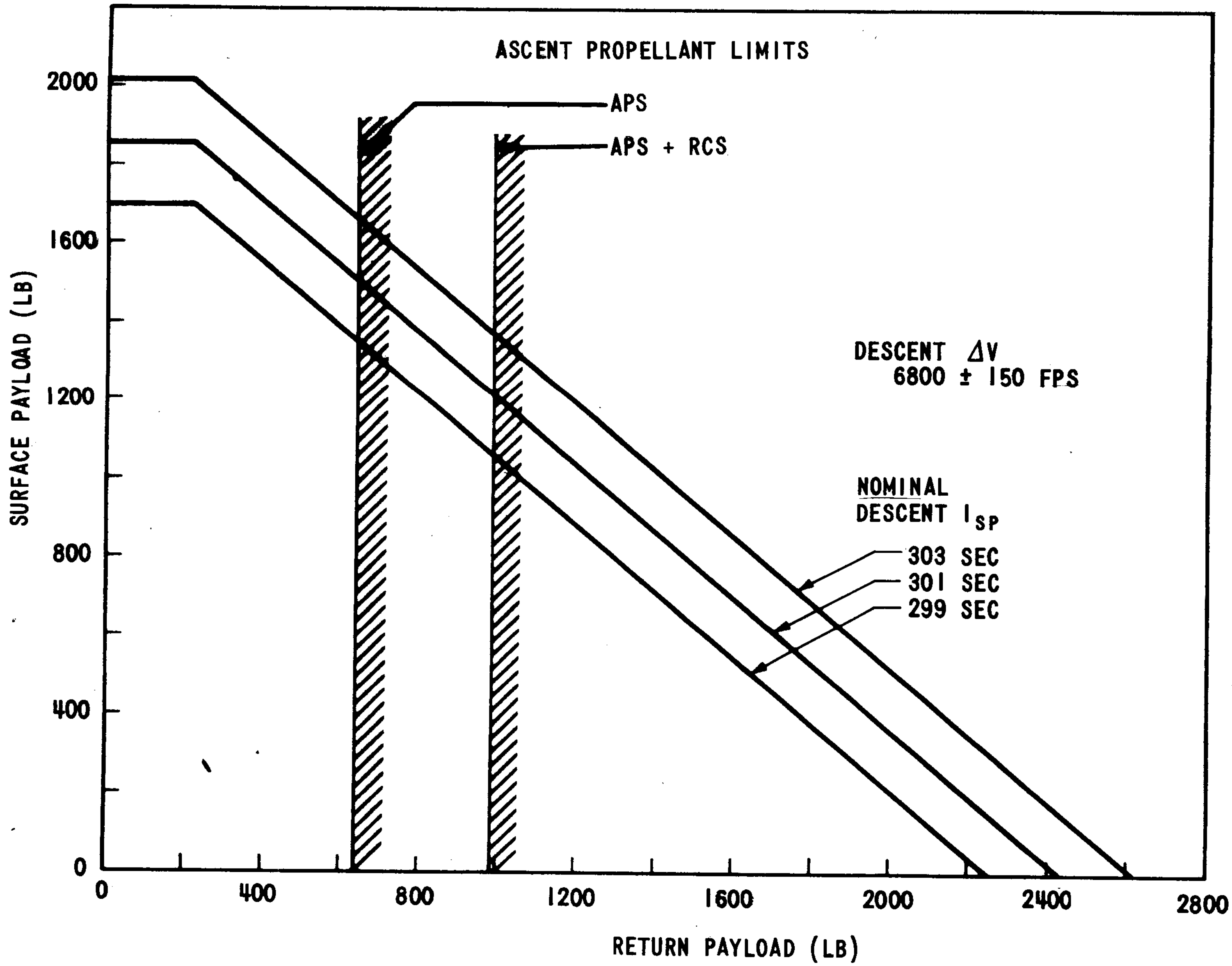

FIGURE 5

LM Weight and Performance

Weights (Pounds)

Descent	
Current Inert Control Weight Less 300 Lbs. Science Payload	4,251*
Landed Payload: Consumables, Habitability, Mobility, Science	1,854
Descent Propellant	17,698
Ascent	
Current Inert Control Weight	4,867*
RCS Propellant	283
Ascent Propellant	4,743
Total LM at Earth Launch	33,696
Return Payload	220

Performance

	Nominal	Dispersion
Descent I_{sp}	301 sec	4 sec
Descent ΔV	6800 fps	150 fps
Ascent I_{sp}	307.5 sec	4.1 sec
Ascent ΔV	6021 fps	10 fps

*Adjusted for modified method of calculating propellant requirements. See Appendix A.

FIGURE 6

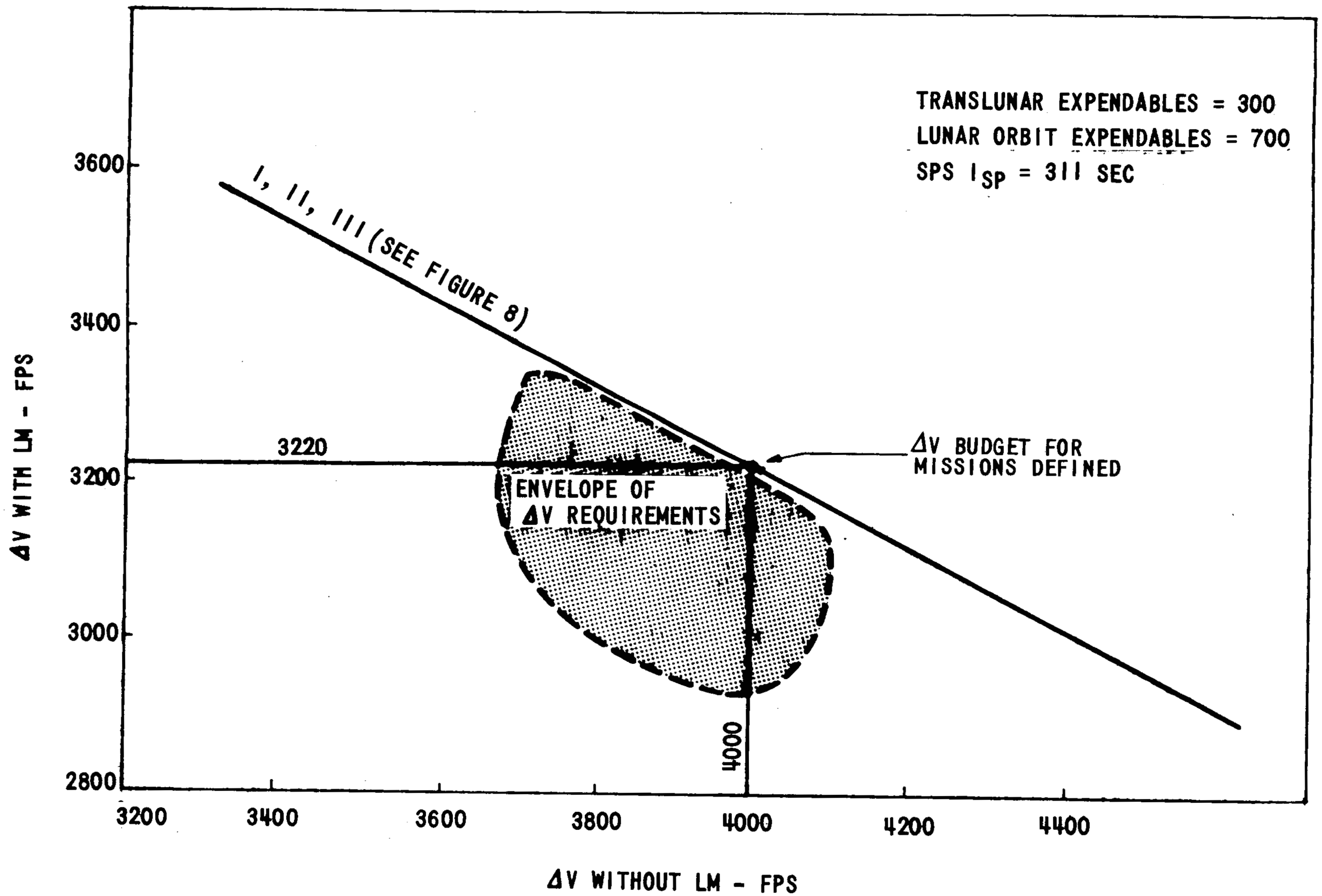

FIGURE 7 - CSM REQUIREMENTS AND CAPABILITY

CSM Weight Models
(Pounds)

	Model I	Model II	Model III
Control Weight CSM (Baseline)	24,800	24,800	24,800
Payload to Lunar Orbit and Returned	1,000	700	1,450
Payload Jettisoned in Lunar Orbit	1,000	2,000	0
SPS Usable Propellant	39,740	39,740	39,740
LM	33,696	33,696	33,696
SLA	4,000	4,000	4,000
	104,236	104,936	103,686

FIGURE 8

Lunar Exploration Sites

Site	Coordinates	Trajectory Type
Abulfeda	14°15'S 14°00'E	Free Return
South of Alexander	38° 4'N 13°29'E	Hybrid 3 impulse (no DPS abort)
Dionysius	2°36'N 17°12'E	Free Return
Aristarchus Plateau	27°28'N 52°16'W	Hybrid 3 impulse (no DPS abort)
Gassendi	17°44'S 40°50'W	Hybrid
Hyginus	7°30'N 7° 6'E	Hybrid
Littrow	22°12'N 29°12'E	Hybrid
Tycho	41°42'S 11°42'W	Hybrid 3 impulse (no DPS abort)

FIGURE 9

BELLCOMM, INC.

Subject: Spacecraft Payload and Mission Profile Changes for Lunar Exploration Missions - Case 310

From: D. R. Anselmo
J. L. Marshall, Jr.

Distribution List

NASA Headquarters

T. A. Keegan/MA-2
W. E. Stoney/MA

MSC

R. L. Berry/FM5
J. Funk/FM8
J. F. Goree/PD5
F. J. Herbert/FA4
C. R. Hicks/FA4
Q. S. Holmes/FM5
C. R. Huss/FM
R. H. Kohrs/PD7
J. P. Loftus/HA
A. J. Meyer/HA
C. H. Perrine/PD
J. R. Sevier/PD12
J. J. Taylor/FM8

MSFC

H. Ledford/R-SE-S

TRW Systems

J. Blahnik
R. J. Gerbracht
P. A. Penzo

GAEC

H. Wagner

Bellcomm, Inc.

A. P. Boysen, Jr.
J. O. Cappellari, Jr.
D. A. Corey
F. El-Baz
D. R. Hagner
W. G. Heffron
N. W. Hinners
T. B. Hoekstra
D. B. James
J. Z. Menard
V. S. Mummert
B. G. Niedfeldt
P. E. Reynolds
F. N. Schmidt
R. Sehgal
D. R. Valley
R. L. Wagner
All Members, Department 2013
Central Files
Department 1024 Files ← COPY TO
Library

Abstract Only

D. A. Chisholm
B. T. Howard
I. M. Ross
J. W. Timko